TURBINE ENGINES FOR HIGH-SPEED FLIGHT

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SUMMARY

Analysis indicates that either a ducted fan or a low-pressure-ratio turbojet engine can be designed for efficient flight at a Mach number of 4, and can be capable of unassisted flight from take-off to the Mach number 4 cruise condition. In comparison with the rocket-boosted ramjet, these engines can provide fundamental advantages in airplane flexibility.

Turbine engines operating over a wide flight-speed range encounter critical problems in cooling and in off-design performance of various engine components, especially the inlet, the compressor, and the exhaust nozzle. Feasible solutions to these problems can be demonstrated at the present time.

Mission studies of a large manned bomber or reconnaissance type airplane indicate a combat radius of near 1200 nautical miles for all cruise flight at Mach 4 and using liquid Methane fuel. A satisfactory engine for this mission is a low pressure ratio turbojet. If diborane is used in the afterburner of the turbojet engine, mission radius can be extended to 2000 nautical miles.

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INTRODUCTION

Present turbojet engines are capable of unassisted flight up to Mach numbers of 2 and it is generally assumed that Mach numbers as high as 3 will soon be possible. This analysis investigates the application of gas turbine engines at a cruise Mach number of 4. In evaluating the potentialities of air-breathing engines at Mach 4 it is important to realize that the principal

competitor in this area is the ramjet engine. In the analysis work done to date it has not been possible to exceed ramjet performance at Mach 4 with any engine cycle considered, and the ramjet is a simpler and lighter engine than any others considered. The advantage of gas-turbine engines lies in the increased airplane flexibility which can be realized with a powerplant capable of unassisted flight from take-off to Mach 4. This is in contrast to the ramjet engine which requires a very large rocket booster to achieve the Mach 4 condition.

In this analysis, turbojet and turbofan engines were considered, with greatest emphases on the turbojet engine. In evaluating the various engine configurations, combat radius of suitable large airplanes was taken as the significant performance criterion. The flight plan in all cases consisted of unassisted take-off and climb to Mach 4, cruise out and back at Mach 4, glide to base at reduced power, and landing with an adequate fuel reserve. The mission is therefore suitable for a large manned bomber or reconnaissance type airplane conducting all of its cruise at a Mach number of 4.

Because of the very large cooling loads imposed by critical engine components such as the bearings, afterburner shell, and exhaust nozzle, a suitable heat sink is a necessary requirement in the engine design. In this analysis it was assumed that this heat sink capacity was provided by using a special fuel. In most cases, liquid Methane was assumed. This fuel would require insulated tanks capable of containing the fuel at -266° F and special pumps and handling techniques would have to be developed. In a few cases,

diborane was assumed as an afterburner fuel only. Even more complex problems would result from the use of this fuel. Detailed analysis of the specialized tankage and fuel systems required for Methane or diborane is beyond the scope of this investigation. However, gross penalties were assessed against the airplane fuel system in an attempt to account for the presence of the special fuels.

No rocket assist was assumed in any of the calculations in order to define the potentialities of unassisted flight with turbine engines. Obviously, some compromises in the direction of rocket assist could be made without great sacrifice in airplane flexibility, for instance, small jato units for take-off or transonic accelerations. Such compromises were not investigated because it is felt that the range potential of the systems involved would fall between that defined here and the range which can be predicted for the fully boosted ramjet engine.

The purpose of the analysis is two-fold. First, to define roughly the potentialities of the weapon systems analyzed, and second to examine the engines defined by the analysis in order to call attention to important problem areas. In this way, research and development effort can be directed toward these areas if the potentialities of the engines are considered worth exploiting.

Cycles

The ramjet operates over the entire Mach number range with an engine pressure ratio of unity. That is, the ramjet relies entirely on ram compression and there is no pressure rise within the engine. Therefore, at low flight speeds where ram pressure is low, ramjet thrust is inadequate. In order to overcome the thrust deficiency of the ramjet, gas turbine engines must produce useful engine pressure ratio at low flight speeds. Figure 1 shows thrust per pound

per second air flow for Mach numbers of 0 to 4 for various values of engine

pressure ratio. The ramjet has an engine pressure ratio of 1 and below Mach

numbers of 2 its thrust decreases rapidly. The turbojet engine considered

in this study has a take-off engine pressure ratio of 1.5 and at Mach 4

its engine pressure ratio falls to .9. Its thrust advantages below Mach

2 are apparent.

The turbofan engine is slightly better than the turbojet at take-off and

slightly inferior at Mach 4. Other turbojet and other turbofan engines could

be examined but these are considered representative for the missions considered.

Figure 2 shows thrust per unit air flow and cycle efficiency as a function

of turbine inlet temperature for a turbojet engine at this Mach 4 flight

condition. Consideration of this figure has led to the choice of an uncooled

turbine and an afterburning configuration in considering a Mach 4 turbojet.

To get adequate thrust and high efficiency at Mach 4 requires high cycle

temperatures, making the afterburning engine look attractive, particularly

since the pressure in the afterburner is almost as high as in the primary

combustor. Consequently, only marginal benefits are realized by going to

elevated turbine inlet temperatures, and these do not justify the complication

involved in designing a cooled turbine.

It is important to note the high level of engine efficiency obtainable;

at Mach 4, over 40 percent for the entire afterburning curve. Therefore, a

Mach number of 4 not only provides important tactical benefits, but it is an

efficient propulsive condition as well.

ENGINE COMPONENTS

Compressors. - Operation at a Mach number of 4 leads to critical

compressor problems and special provisions were taken in synthesizing

the design so as to alleviate these problems as far as possible. Figure 3

represents a typical compressor map of a turbojet engine which might be
designed to fly to a Mach number of 2.5 or 3. An operating line for a
constant speed engine is included in the figure with points representing
take-off and flight at various Mach numbers in the isothermal atmosphere.
The .9 Mach number condition is located at 100 percent equivalent speed.
If engine operation is pushed to Mach 4 with this compressor, rotating
stall and low compressor efficiency will be encountered as can be appreciated
from the location of the Mach 4 operating point. Also this type of operation
will require a large variation in inlet corrected flow and will require a
variable inlet throat for satisfactory inlet operation . In synthesizing
Mach 4 performance for this study a different type of compressor and operating
line was assumed. Four changes were made to improve the engine operation up
to Mach 4. First, compressor pressure ratio was lowered to a value of 2.4
at take-off. This results in a less sensitive compressor with a broader
plateau of high efficiencies and flatter speed lines. Second, the design
compressor tip speed was lowered from 1,050 to 750 ft/sec. This decreased
blade Mach numbers and again makes the compressor less sensitive with a
broad efficiency plateau. It also allows aerodynamic overspeed capacity
for low flight speed operation. Third, take-off was taken as maximum engine
temperature ratio instead of Mach .9 in the stratosphere so as to reduce the
range of compressor operation required in flying to Mach 4. This results in
a derated turbine inlet temperature with some loss in thrust at Mach 0.9, but
the airplane calculations have indicated this is not a critical thrust region
in any case. The fourth change was to depart from constant speed operation
and increase mechanical speed up to Mach 3.3, with constant mechanical speed
above this point. This results in Mach 4 operation at 78 percent equivalent
speed for an engine with a take-off point at 115 percent equivalent speed.

The results of these changes are shown in figure 4 which is the compressor map and operating line synthesized for the turbojet engine studied in this analysis. The map was obtained by analytically stacking data taken on a single stage compressor. The take-off point is located at 115 percent equivalent speed and up to a Mach number of 3.3, compressor flow is determined by the limiting loading line of the turbine. Turbine outlet Mach number was assumed to be .6 along this part of the operating line. Above Mach 3.3 the compressor runs at a constant mechanical tip speed of 1,020 feet/sec. The Mach 4 point is not near rotating stall because it is located far from the surge line. Front stage blade angle of attack at Mach 4 is 4 degrees less than at take-off. Thus satisfactory compressor operation is achieved at Mach 4 in a stall free region. Pressure ratio at Mach 4 is only 1.25 and operation is essentially like a ramjet with over 90 percent of the total fuel being added in the afterburner.

The compressor operation described above applies to the turbojet engine. Similar concepts were applied in synthesizing the fan and compressor performance for the turbofan engine.

Burners. - Gaseous Methane fuel has combustion characteristics similar to vaporized JP fuel in the temperature and pressure ranges to which it will be subjected in Mach 4 engines. Because of the low compressor pressure ratio necessary in the turbojet engine, primary combustor size will be large. In fact, a primary burner velocity of 250 ft/sec results in a burner frontal area $2\frac{1}{2}$ times compressor frontal area. Although it is desirable to keep burner area small to conserve engine weight, very large burner areas can be accommodated if necessary because the inlet and the outlet are by far the largest components of the engine. A maximum afterburner velocity of 400 feet per second results in an afterburner area $2\frac{1}{2}$ times compressor frontal area. The co-burner and

afterburner in the turbofan engine are similar to the turbojet afterburner.
The primary combustor in the turbofan is more easily kept to a small size
because less than half the total air flow passes through it.

Turbine. - Because of the low compressor pressure ratio in the turbojet
engine, density of the gas at the turbine exit is low. Therefore, to keep
the size and weight to a minimum it is necessary to design a high flow
turbine. This leads to low radius ratios and high turbine exit Mach numbers.
Figure 5 presents the variation of turbine size with radius ratio for a Mach
4 turbojet engine, assuming a turbine exit axial Mach number of 0.6. To
limit turbine diameter to 116 percent of compressor diameter it is necessary
to reduce radius ratio to 0.5. This results in a low hub speed, but even a
single stage turbine has adequate work capacity for a low pressure ratio
turbojet of the type considered. Low radius ratio leads to long blades and
low work output leads to small blade camber; so that the principal problem
connected with this turbine is structural. Long blades with low camber are
prone to vibrational problems and new turbine design techniques may be needed.
One approach would be to design blades with tapered chords to obtain more
desirable structural shapes. This will necessitate an aerodynamic design
approach similar to the free airfoil theory used in compressor design rather
than the channel flow theories adhered to by turbine designers in the past.

Problems in the turbofan turbine are almost the opposite because work
output rather than size is the primary consideration. The work output per
pound of fluid passing through the turbine is high because the turbine must
drive both a compressor and a fan with less than half the fan flow going
through the turbine. Also, with less than half the fan air flow going through
the turbine, the turbine will be smaller in diameter than the fan, and the
turbine tip speeds are limited by the speed and stress capabilities of the fan.

As a result of this high work output and low speed of the turbine, multi-stage turbines are unavoidable, but it appears that conventional design techniques will be adequate.

Inlets and outlets. - At off-design conditions, the largest losses in the propulsion system are those associated with the inlet and outlet. Critical flow at Mach 4 requires a large lip diameter, usually one and one-half to two times compressor diameter. This lip size is excessive at transonic conditions and operation in the transonic region incurs large additive drag losses.

Similarly, a Mach 4 engine will require a nozzle designed for a pressure ratio of 40 to 80. At transonic speeds the nozzle exit area is excessive and over-expansion losses result. Because afterburning was shown to be essential, variable nozzle throat area is required for proper engine control.

The inlet and exit problems can be approached in two ways. Either simple fixed components can be designed and large transonic losses accepted, or complex variable geometry components can be developed with reduction in off-design losses. The variable-geometry components will undoubtedly be heavier, so some sacrifice in engine weight is involved when variable components are considered. In between these two approaches there may be an attractive middle course, if it proves feasible to achieve some of the benefits of variable geometry with a minimum of complexity. For instance, wind tunnel tests at Mach 1.5 to 2.0 have indicated that effective cones of stagnant air can be established behind the oblique shock structure of a probe or pointed rod attached ahead of a blunt body. If this can be successfully exploited forward translation of the first cone of a two-cone inlet at low Mach numbers will result in cone pressures similar to those which would be expected on a single cone inlet and large savings over the two cone additive drag will result. Another means which might be employed to reduce additive drag without too much complication, would be to treat it like any other flow deflection in the airplane design and compensate for it by applying the transonic area rule. Wind tunnel tests will be necessary to evaluate the validity of these ideas.

In addition to the inlet losses discussed above, further losses can result if subcritical or supercritical inlet operation is required to match the engine airflow schedule. Also, inlet instability can result if inlet operation becomes excessively subcritical. It is therefore important to examine the inlet-engine airflow match over the entire supersonic speed range. Figure 6 shows inlet critical flow and engine required flow for flight Mach numbers of 1 to 4. As was mentioned in discussing the compressor, engine operation was deliberately synthesized to match inlet flow as far as possible and figure 6 shows the extent to which this was successful. Maximum differences are 4 percent supercritical and 10 percent subcritical. With this match it would be easy to accomplish critical inlet operation over the entire range by bypassing some air around the engine as required by the inlet schedule.

Experimental data on conical inlets indicate that the critical weight flow variation is flatter than that predicted by theoretical calculations. An experimental inlet then will match the engine weight flow schedule even more closely than indicated in figure 6.

Turbofan airflow characteristics can be treated similarly in order to match critical inlet flow as nearly as possible.

Because inlet throat area is fixed by the Mach 4 condition, and because sharp lipped inlets will be used, take-off doors will have to be provided.

Cooling. - Because the engine is immersed in an environment of 1100° to 1200° F, critical cooling problems will be encountered. Most important are probably accessories, bearings, afterburner shell, and exhaust nozzle. Several approaches to these problems have been proposed, and one practical solution appears to be through the use of compressor bleed air cooled in one or two fuel-to-air heat exchangers. This requires a satisfactory heat sink capacity in the fuel, and has led to the choice of refrigerated liquid methane for this study. Because the Mach 4 condition is most critical, and because most of the

engine fuel flow is going into the afterburner at Mach 4, only afterburner fuel is assumed to be diverted through the heat exchangers. In order to extend the radius capability of the airplane configurations studied, some calculations were made assuming diborane for the afterburner fuel and methane for the primary combustor fuel. The heat sink capacities of methane and diborane afterburner fuels are compared in figure 7, which show total heat capacity and engine heat sink requirements. It was assumed that both fuels were stored in the tanks at their freezing point. Because diborane decomposes at elevated temperatures it was assumed that 150° F was the highest temperature to which it could be heated. And because methane is more stable at elevated temperatures, a maximum possible temperature of 1000° F was assumed for it.

It is apparent from figure 7 that the heat sink capacity of methane is very much superior to that for diborane.

The cooling load for a single engine with a compressor diameter of 34 inches was estimated to be 1590 Btu's per second at a Mach number of 4 and 85,000 feet. This represented 63 percent of the heat sink capacity of the diborane afterburner fuel and leaves only 37 percent for other heat sources which are always present, such as heat leaks, friction, and so forth. The applicability of diborane to the Mach 4 condition is therefore subject to some question in view of its limited heat sink capacity. If more detailed design studies reveal that the diborane fuel is inadequate at Mach 4 from a heat sink standpoint, it would be possible to design a refrigerating system to absorb all or some of the engine cooling load. However, this would add weight and complexity to the propulsion system which were not assessed in this study.

The cooling loads of the turbofan engine are higher than the turbojet because of the necessity of cooling the duct separating the two exhaust streams. Application of diborane to this engine was not considered.

Engines

Consideration of the component problems discussed in the previous section led to the selection of specific designs for the analysis of off-design performance and mission capabilities. The particular designs were not chosen because they are considered the optimum engines that can be conceived but they should be representative, and they served to limit the scope of the investigation.

Turbofan. - Figure 8 shows a layout of the turbofan engine considered in the analysis. The design consists of a two-stage fan directly connected to a six-stage compressor and a three-stage turbine. Separate exhaust ducts were assumed for the primary and by-pass gas flows. A single exhaust duct might be a more practical configuration but will necessitate windmilling operation of the fan at Mach 4, and the uncertainties of estimating the performance of a windmilling component were avoided in this analysis by using separate ducts. The inner duct exhausts through a fixed-area nozzle and the outer duct through a variable-area nozzle. Turbine-inlet temperature is 2100° R which will allow an uncooled turbine. Exhaust temperature of the primary duct is 3500° R and the by-pass duct exhaust temperature was 4000° R for climb and 3500° R for Mach 4 cruise. All burners are designed to operate on gaseous Methane fuel. Air is bled from the secondary duct and cooled in a heat exchanger. Part of the bleed air is cooled further in a second heat exchanger and is then diverted to the engine bearings and bearing housings. The remaining bleed air is used to cool the afterburner shell, the convergent variable exhaust nozzle and duct and nozzle separating the two exhaust streams. Uncooled bleed air is introduced just aft of the throat of the ejector-type variable-area nozzle. At take-off, fan pressure ratio is 1.7, compressor pressure ratio is 3, and 60 percent of total air flow is by-passed through the secondary duct. At Mach 4 fan pressure ratio is 1.1, compressor pressure ratio is 1.4 and by-pass air flow is 70 percent of total.

Several inlets were considered in this study. The inlet shown in figure 3 is a two-cone symmetric design with cone angles of 20° and 35°. A bleed slot is provided at the shoulder of the second cone to aid in turning the air into the subsonic duct. Also, by controlling the amount of bleed through the slot the inlet can be kept critical over the entire supersonic climb path. The air bled through this slot is ducted to the rear of the engine and exhausted through slots in the divergent section of the nozzle. At low flight speeds the nozzle would tend to overexpand the flow and the introduction of this shoulder bleed air near the nozzle exit will promote separation and raise base pressures. Because of the strength of the terminal shock at Mach 4, no diffusion of the air is necessary in the subsonic duct and duct length can be kept to a minimum.

Turbojet. - The turbojet layout is presented in figure 9. It consists of a three-stage compressor and a single-stage uncooled turbine, with a maximum inlet total temperature of 1900° R. Afterburner temperature was 4000° R for climb and 3500° R for cruise. A variable-area exhaust nozzle throat was assumed and the burners in figure 4 were designed for gaseous Methane fuel. Maximum primary burner velocity was 250 feet per second and the maximum afterburner velocity was 400 feet per second. These velocities were similar to those assumed in the turbofan engine. About 10 percent of the compressor-exit airflow is diverted for cooling and nozzle secondary flow. Compressor pressure ratio is 2.4 at sea level and 1.25 at Mach 4. The inlet shown in figure 4 is similar to the one described for the turbofan.

Mission Studies

The mission on which the analysis was based assumed a manned bomber or reconnaissance type airplane. Gross weight was 250,000 pounds, payload weight was 10,000 pounds, and fixed-equipment weight was 20,000 pounds. A highly swept delta wing with an aspect ratio of 1.5 and thickness ratio of 0.025 was assumed, together with a long, thin fuselage with a fineness ratio of 20. The analysis

of a particular configuration type is necessary in order to obtain meaningful
results, but no particular claim is made that the airplane assumed is optimum for
the mission. Wing, fuselage, and engine size were all varied with design altitude
in order to obtain a series of airplanes, each designed for the particular mission
under consideration. In all mission studies, the results are critically dependent
on the assumptions involved, and are useful primarily for the comparison of differ-
ent engines, fuels, and other propulsion-system variables. The absolute values
of the results are less significant and should be interpreted only in defining
capabilities of the systems analyzed in a general sense. More detailed design
studies of the airplanes and engines involved will be necessary to obtain definite
values of airplane radius.

Inlets. - In order to bracket the inlet problems to some extent, three inlet
configurations were investigated. The first was a 25° half-angle cone of conven-
tional design with fixed geometry at all conditions. This inlet gave a pressure
recovery at Mach 4 of .3 and resulted in large engine sizes and some increase in
specific fuel consumption at Mach 4. However, at off-design conditions the inlet
was quite satisfactory, additive drag was low and the large engine sizes required
behind it resulted in adequate thrust margins over the climb path.

The second inlet investigated was a two-cone, three-dimensional design with
cone angles of 20° and 35°. Mach-4 recovery was .44. The geometry of this inlet
was assumed to be completely variable so that no additive drag was charged at any
off-design flight conditions. In actual practice this would require a complicated
and heavy mechanical design. It could probably be accomplished more easily in a
two-dimensional design, but even here weight penalties would result. In evaluat-
ing the variable inlet it was further assumed that design-point nozzle perform-
ance could be obtained over the entire flight path. This is in contrast to the
nozzle performance assumed for the other engine calculations where nozzle thrust

coefficient was varied with Mach number to take account of over-expansion losses. No extra weight or drag was charged to the variable-inlet design. Thus, the two-cone completely variable inlet represents the maximum possible benefit which can be expected from variable-inlet and exit geometry, since all benefits were assumed for it and no added penalties were assessed against it.

The third inlet investigated was similar to the two-cone ideal inlet just discussed, but the standard nozzle performance was assumed and some additive drag was assessed against it at off-design conditions. It is intended to represent the limits of off-design inlet performance which can be achieved with only slight complexity due to variable geometry.

For purposes of identification, the three inlets will be described as the single-cone inlet, the ideal two-cone inlet, and the minimum-drag two-cone inlet. Thrust margins in the transonic region were smallest with the minimum-drag two-cone inlet and are presented in figure 10 along with the climb path. The engine thrust plotted in figure 10 is the net thrust minus inlet and nacelle drag, and the drag is that associated with the airplane alone without including engine and nacelle drags. Because of high engine and airplane drags in the transonic region, thrust margin is smallest at a Mach number of 1.5 where thrust is 25 percent greater than drag. The other two inlets had larger thrust margins and the higher target altitudes, which required larger engines and also had larger thrust margins. For the configuration represented in figure 10, total fuel consumed during climb was 67,000 pounds or 58 percent of the total take-off fuel load. Range covered during climb was 330 nautical miles, and time to climb to the initial cruise altitude of 86,400 feet was 20 minutes.

The radius capabilities of turbojet engines equipped with the three different inlets are presented in figure 11 as a function of target altitude. Maximum radius of 1350 nautical miles was obtained with the ideal two-cone inlet at a target

altitude of 88,000 feet. Best radius for the single-cone inlet was 1050 nautical miles and the minimum-drag two-cone inlet gave performance between the other two but well up toward the ideal inlet. Target altitudes lower than those shown for each curve were not investigated, because thrust margins either at take-off or in the transonic region were considered inadequate. Increases in target altitude affect all three inlets similarly and it appears possible to design for target altitudes in excess of 102,000 feet, although radii of less than 1000 nautical miles would have to be accepted at these altitudes.

Engines. - Figure 12 compares the radius capabilities of the turbojet and turbofan engines with two-cone minimum-drag inlets. No significant differences in performance were revealed by this investigation. It should be emphasized that this conclusion is valid for the engines of this analysis. Other turbofans or turbojets could compare differently. These particular designs were chosen as representative of Mach-4 engines.

Fuels. - To increase the radius capability described for the turbojet engine in figure 12, the use of diborane fuel in the engine afterburner was investigated. Substitution of diborane in the afterburner only is very nearly as good as substitution of diborane in both primary burner and afterburner because over 90 percent of the engine fuel flow goes into the afterburner at the Mach-4 condition. The results of using diborane in a turbojet engine with a two-cone minimum-drag inlet are presented in figure 13. At a target altitude of 90,000 feet a radius of 2050 nautical miles is predicted using diborane fuel. The methane curve of figure 13 is repeated from figure 10 and it can be seen that the use of diborane increased radius about 70 to 90 percent at all altitudes. The calculations necessary to obtain figure 11 took account of the decrease in apparent heating value of the diborane fuel due to vaporization of boric oxide in the afterburner. This effect is rather serious at 4000° R but not so serious below 3500° R. For this reason,

afterburner temperature was reduced to 3500° R during boost and 3300° R during cruise in completing the diborane calculations. It should be emphasized that use of diborane will incur very serious storage and handling problems. The data of figure 13 reflect the potential of the fuel, but more detailed studies are necessary to define its practicability. In calculating required fuel-tank volume, a density of 25.5 pounds per cubic foot was used for both methane and diborane fuels.

EFFECT OF ENGINE PRESSURE RATIO

Fig. 1.

If another grid must be used, maintain 5 x 7 proportion.

NACA--C--854b (7-24-56)

EFFECT OF TURBINE INLET TEMPERATURE

TURBOJET ENGINE
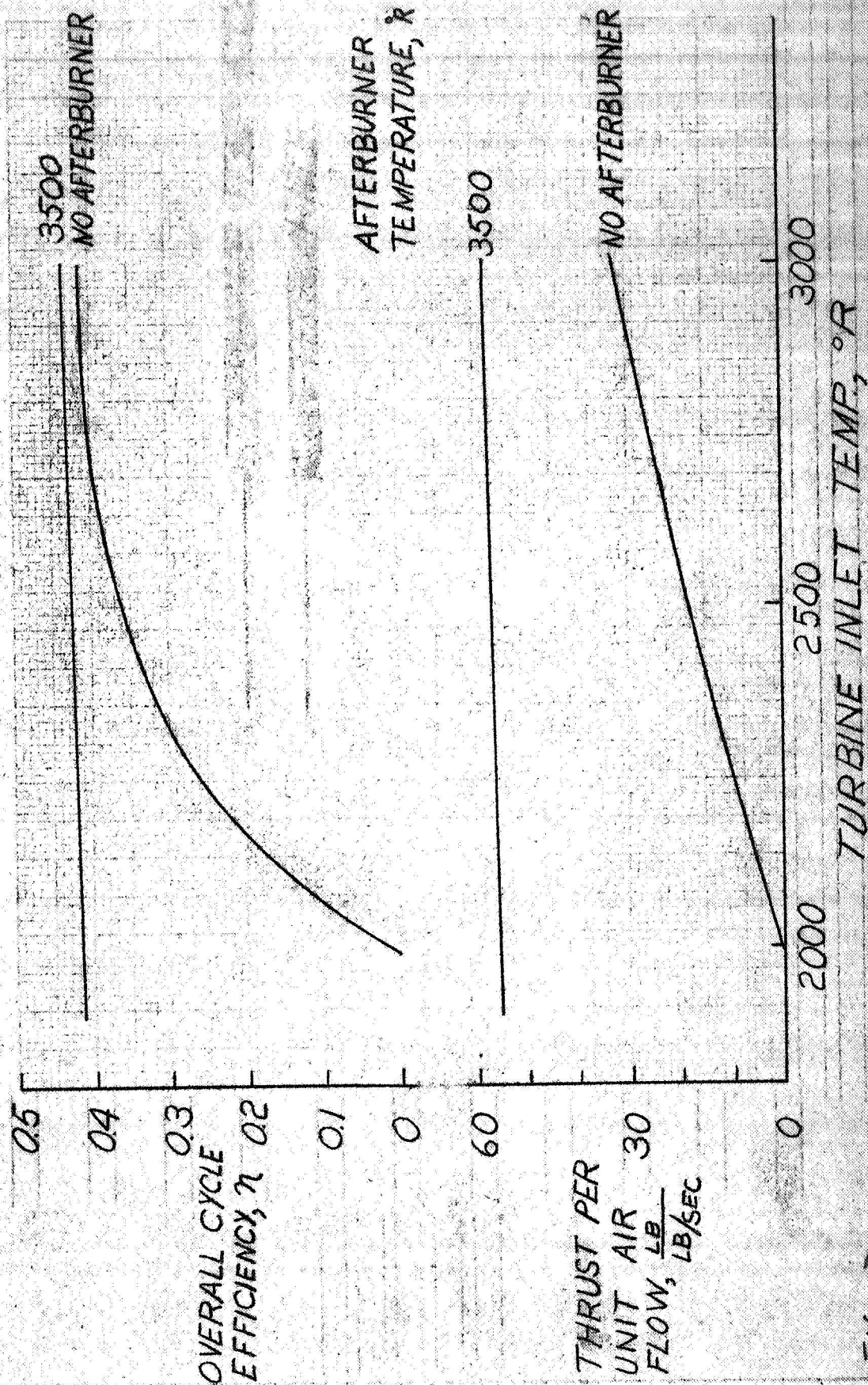
Fig. 2.

5 STAGE COMPRESSOR MAP

$$(U_t/\sqrt{\theta})_{DES} = 1050 \ FT/_{SEC}$$
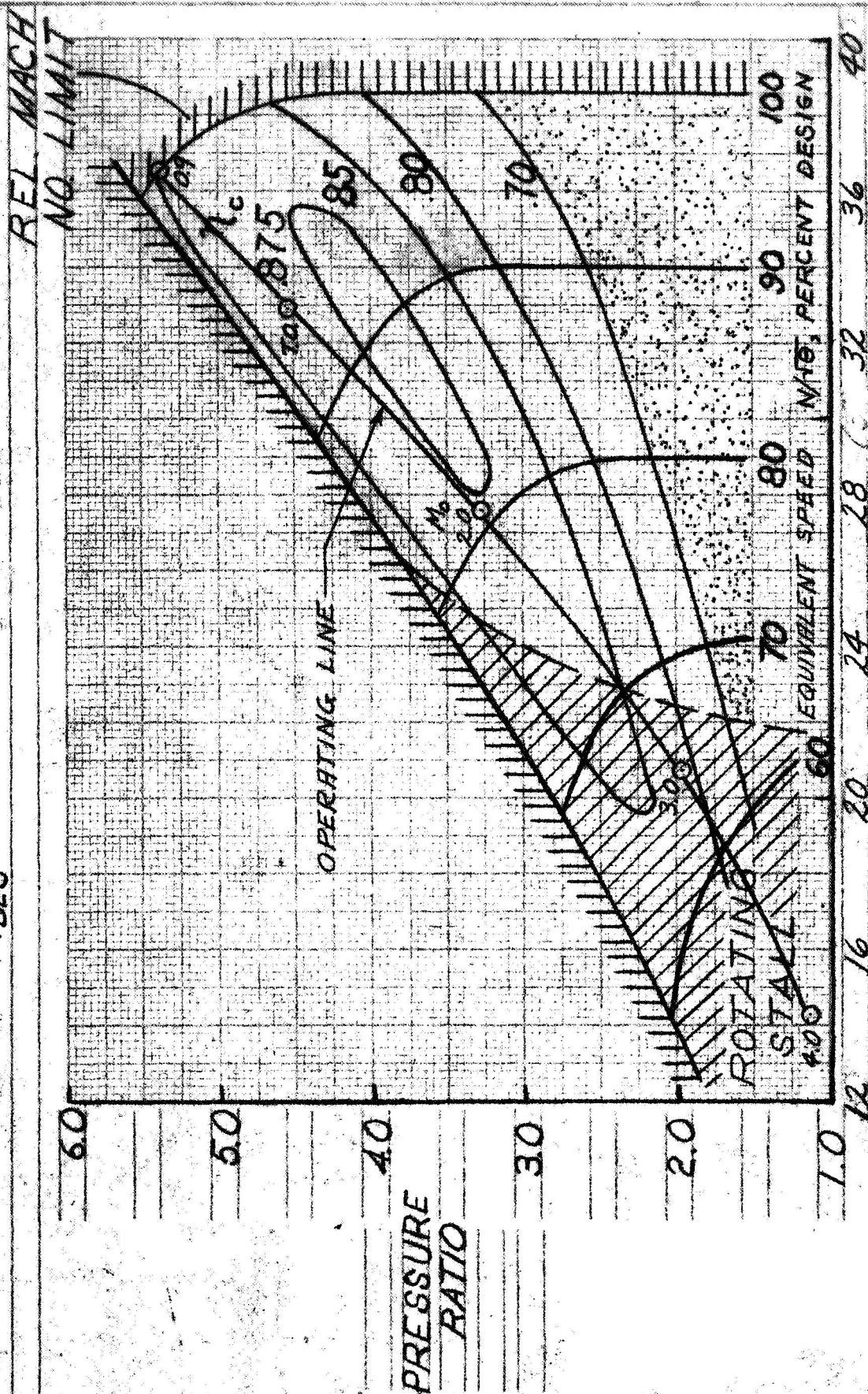
FIG. 3.

If another grid must be used, maintain 5 x 7 proportion.

Brief ordinate title (horizontal)

3 STAGE COMPRESSOR MAP
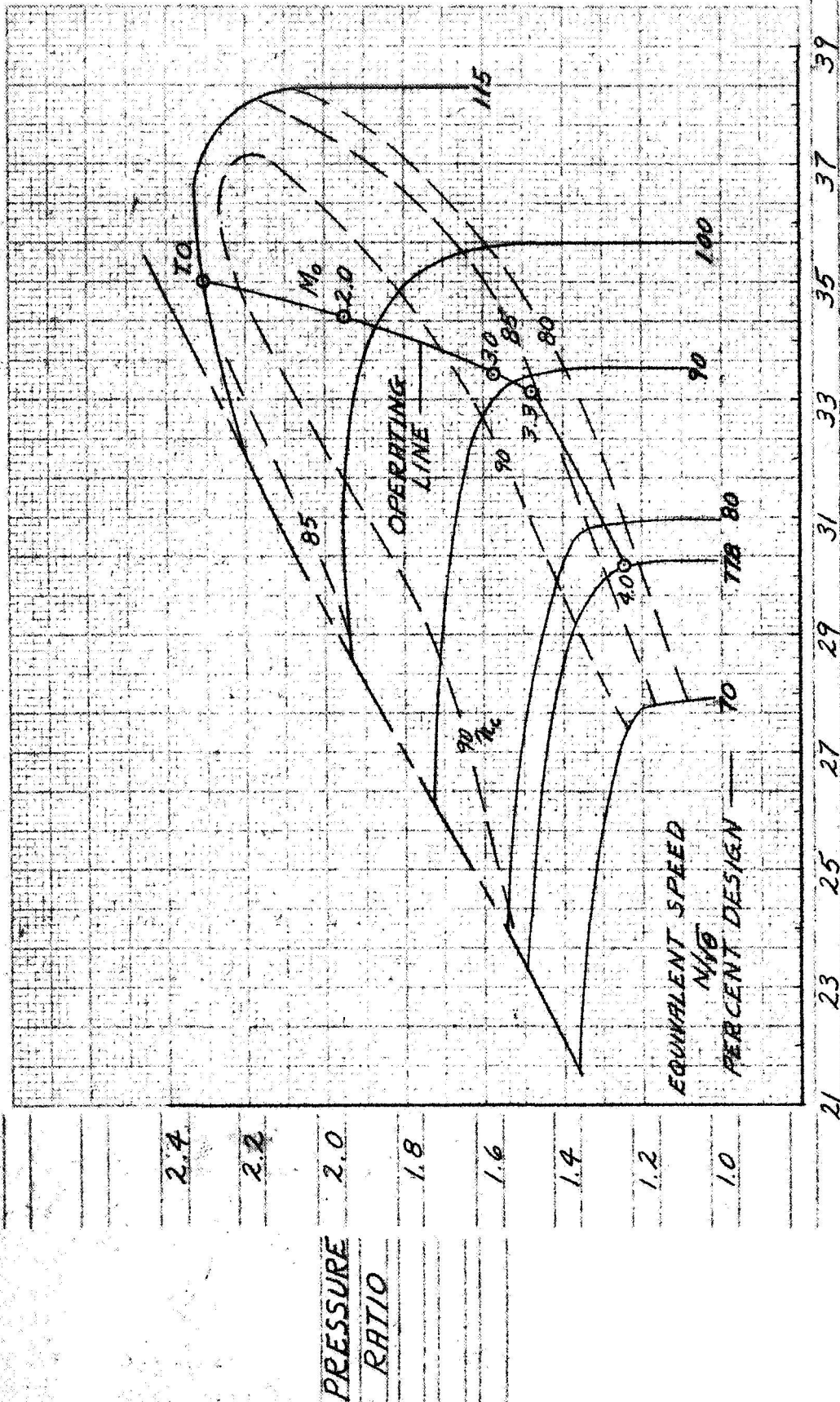
$$(\sqrt{4/\theta})_{DES.} = 750 \text{ FT/SEC.}$$

EQUIVALENT SPEED
$N/\sqrt{\theta}$
PER CENT DESIGN

PRESSURE RATIO

CORRECTED AIR FLOW, $\frac{\#/SEC.}{FT^2}$

FIG. 4.

NACA–C–854b (7-24-56)

TURBINE SIZE

TURBOJET ENGINE

$$\frac{TURBINE\ DIA.}{COMPRESSOR\ DIA.}$$

RADIUS RATIO

3.0
2.0
1.0
0

0   0.2   0.4   0.6   0.8   1.0

FIG. 5.

INLET AND COMPRESSOR FLOW CAPACITY
TURBOJET ENGINE

FLOW CAPACITY 2 CONE INLET

THEORETICAL INLET CRITICAL FLOW

ENGINE FLOW

COMPRESSOR
CORRECTED
AIR FLOW,
$\frac{\#/sec}{FT^2}$

40

30

20

10

0.10

2.0   30   4.0

FLIGHT MACH NUMBER

FIG. 6

SLIDE ORIGINAL GUIDE SHEET
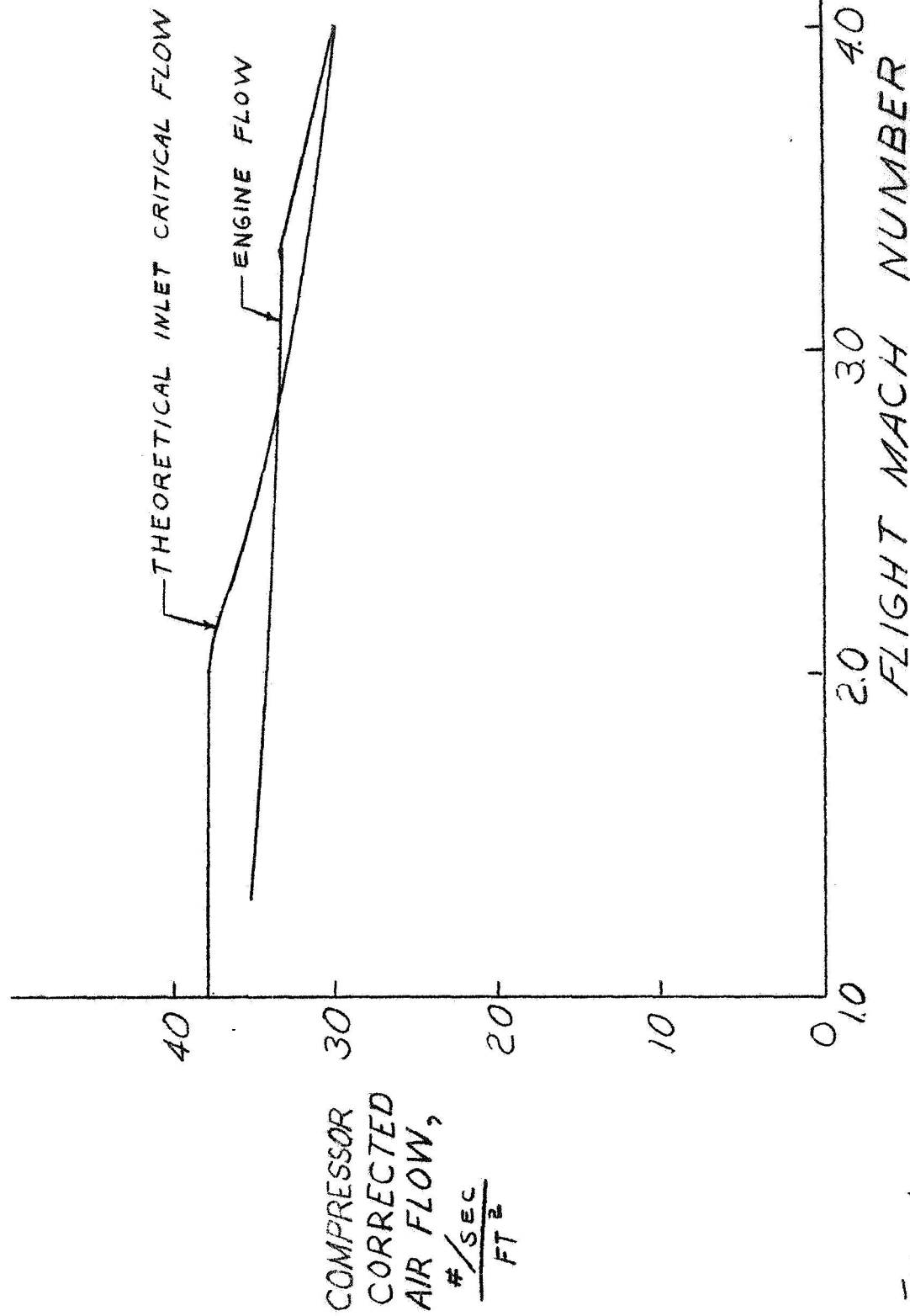
AFTERBURNER FUEL HEAT SINK CAPACITY

MACH No. 4   85,000 FT

TURBOJET ENGINE   COMPRESSOR DIA. 3½ IN.
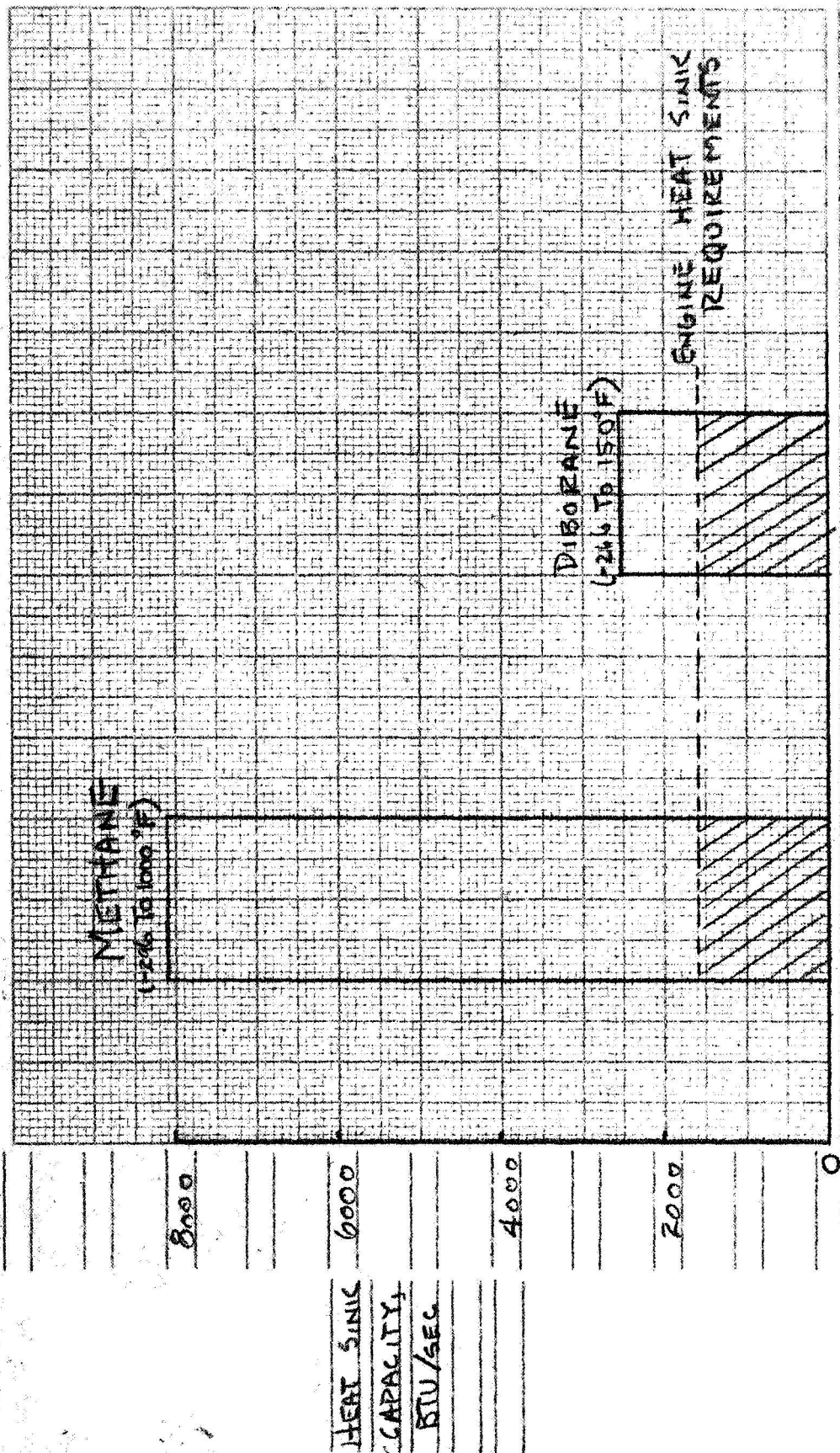
METHANE
(-246 TO 1000 °F)

DIBORANE
(-246 TO 150 °F)

ENGINE HEAT SINK
REQUIREMENTS

HEAT SINK
CAPACITY,
BTU/SEC.

8000

6000

4000

2000

0

Brief ordinate scale title (horizontal)

Fig. 7.

NACA-C-854b (7-24-56)

If another grid must be used, maintain 5 x 7 proportion.

TURBOFAN ENGINE

FIG. 6.

TURBOJET ENGINE

FIG. 4.
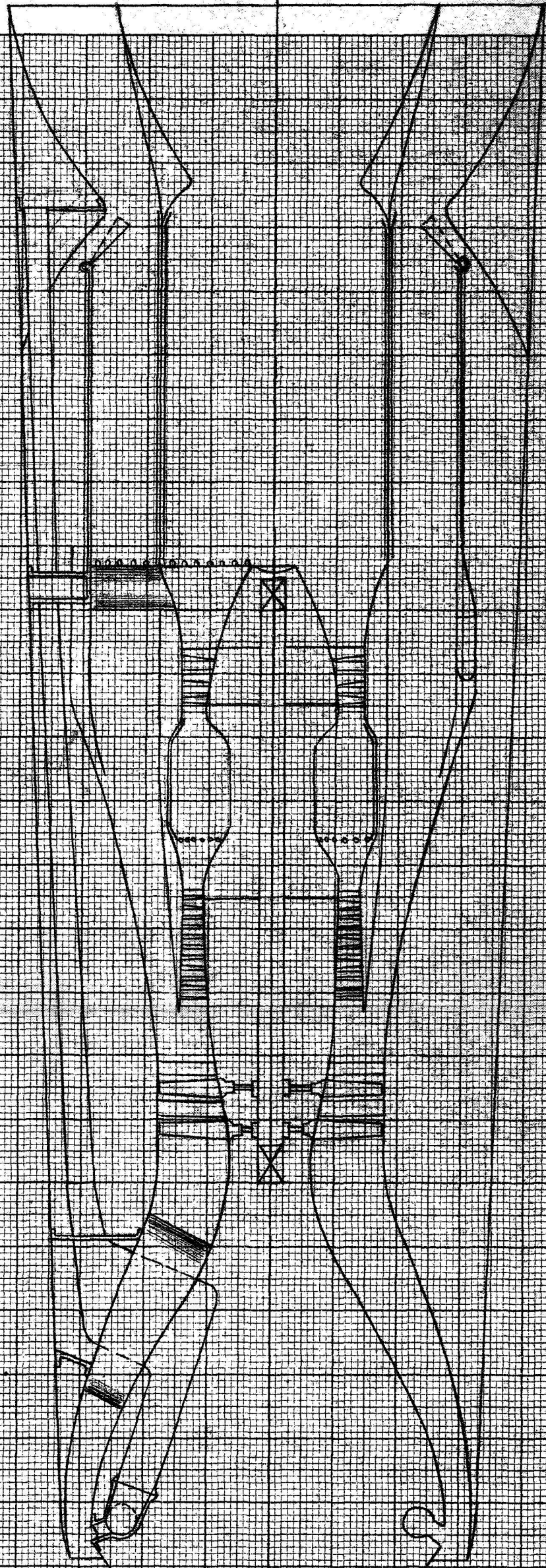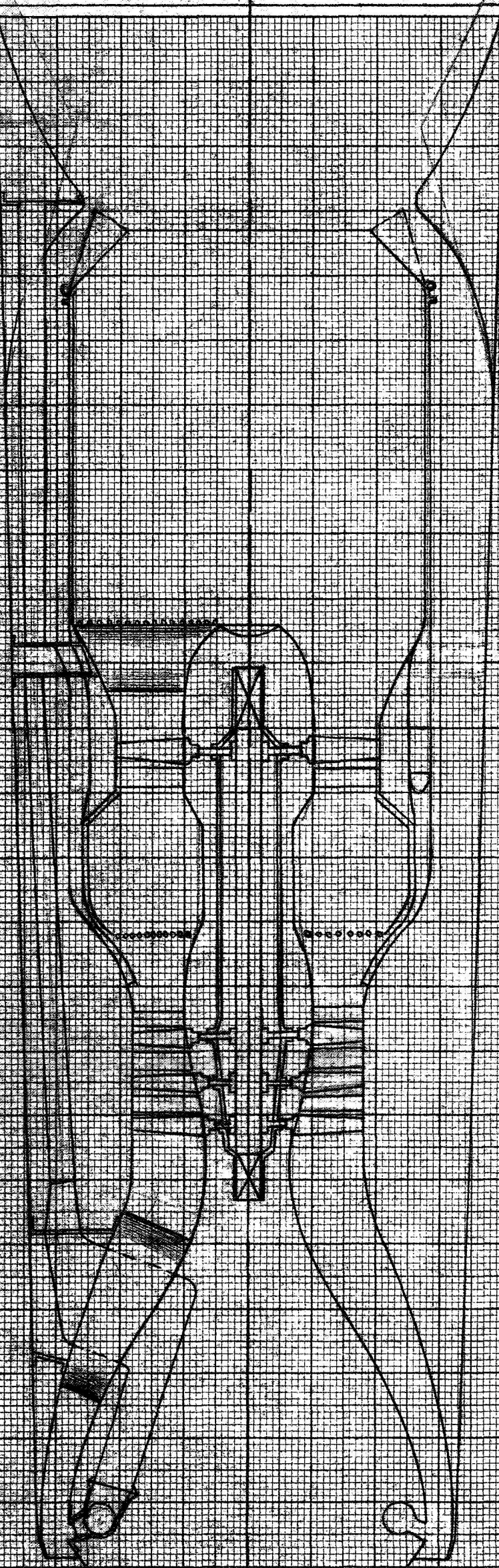
CLIMB PERFORMANCE

SLIDE ORIGINAL GUIDE SHEET

TURBOJET ENGINE

TWO CONE MIN. DRAG INLET

TARGET ALTITUDE, 90,000 FT
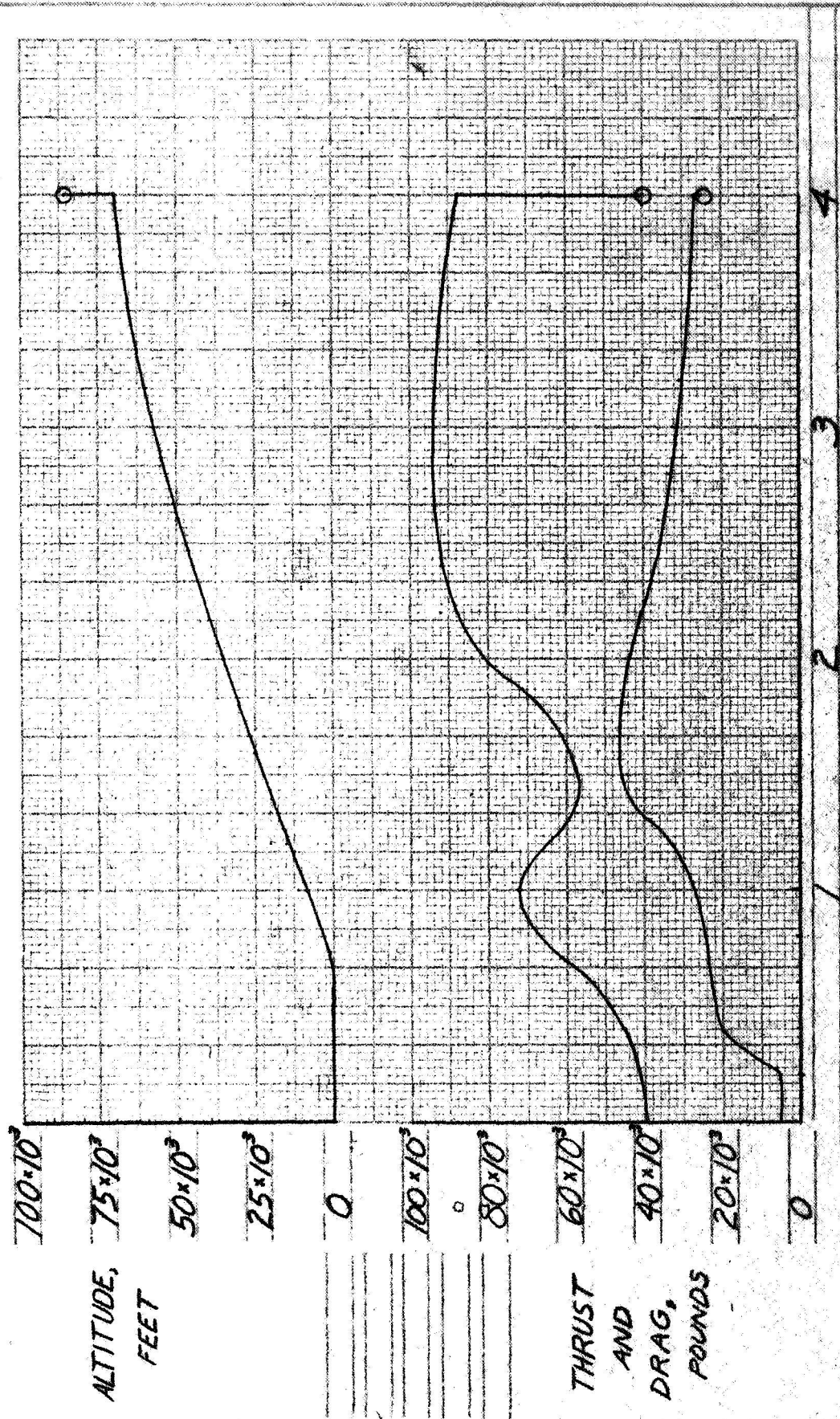
FIG. 1.10.

SLIDE ORIGINAL GUIDE SHEET

AIRPLANE RADIUS WITH DIFFERENT INLETS

MACH No. 4

TURBOJET ENGINES   METHANE FUEL
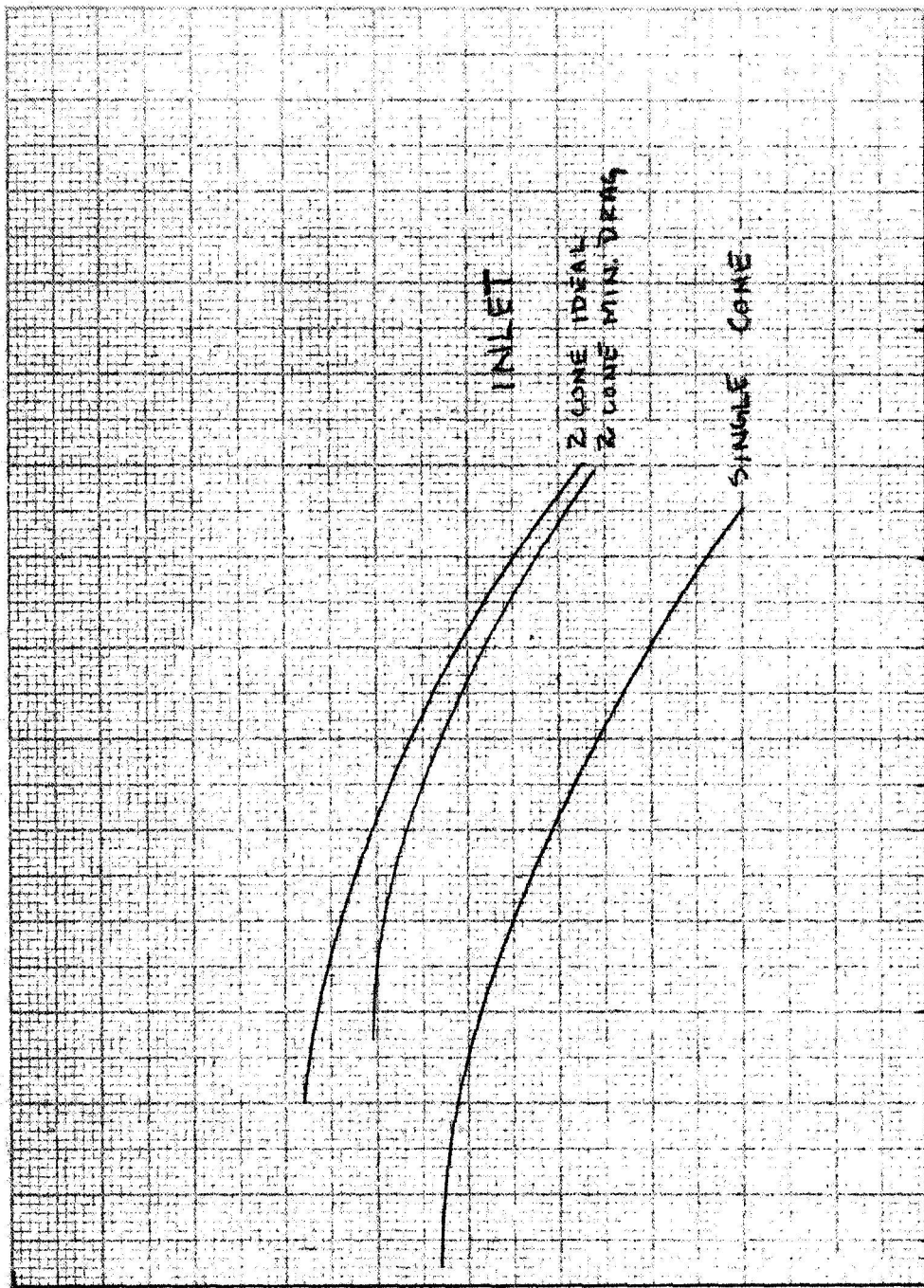
RADIUS,
NAUT. MI.

TARGET ALT, FT.

Fig. 11.

If another grid must be used, maintain 5 x 7 proportion.

Brief ordinate scale title (horizontal)

SLIDE ORIGINAL GUIDE SHEET

Airplane Radius with Different Engines

Mach No. 4

Methane Fuel

2 Cone, Min. Drag Inlet

RADIUS, 1000 NAUT. MI.

Brief ordinate scale title (horizontal)

Fig. 12.

NACA-C-854b (7-24-56)

SLIDE ORIGINAL GUIDE SHEET
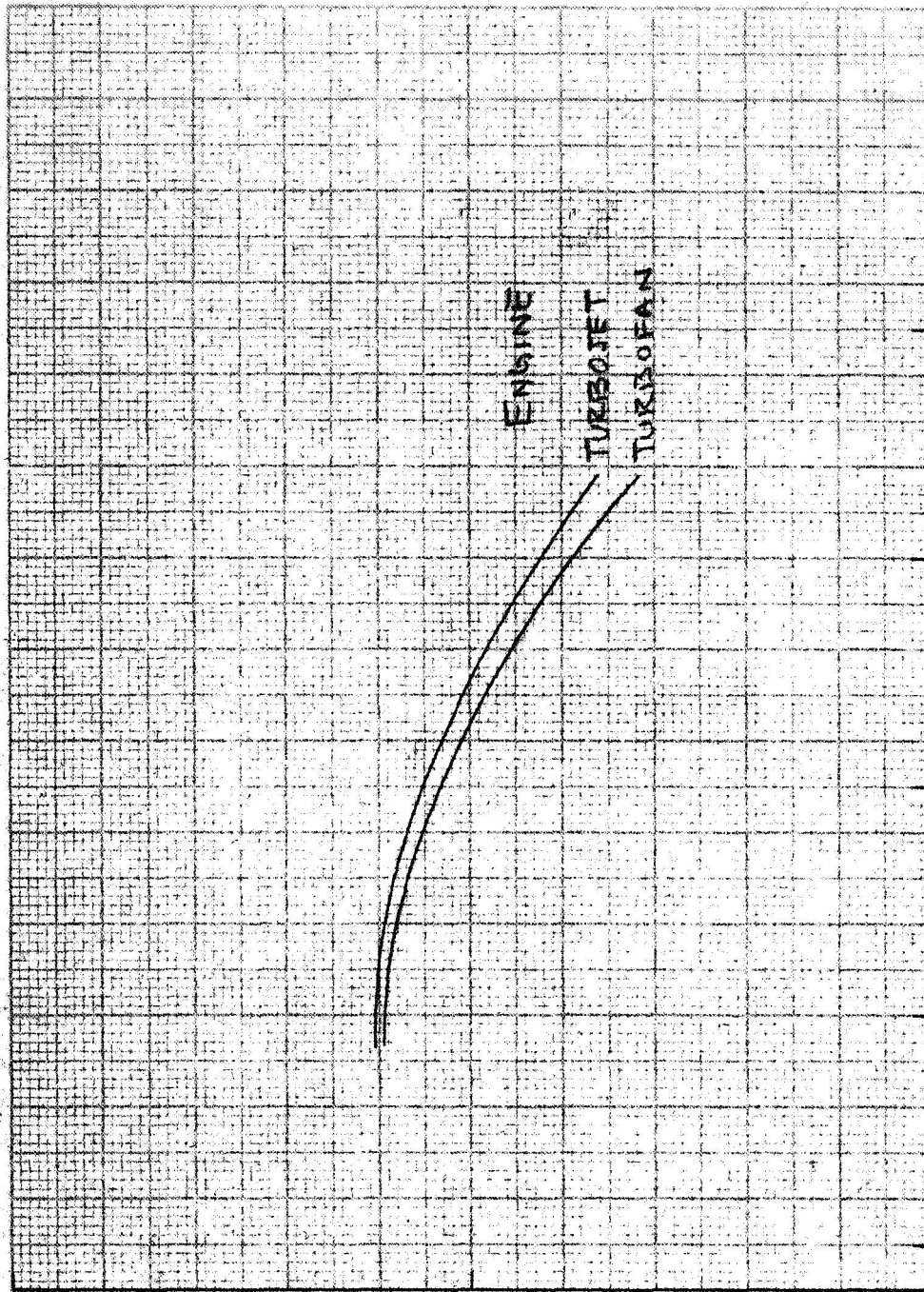
AIRPLANE RADIUS WITH DIFFERENT AFTERBURNER FUELS

Mach No 4

2 CONE, MIN DRAG INLET

TURBOJET ENGINES

AFTERBURNER
FUEL

DIBORANE

METHANE

RADIUS
NAUT MI.

2000

1000

0

85   90   95   100   105   × 10³

TARGET ALT., FT.

Fig. 13.

Brief ordinate scale title (horizontal)

If another grid must be used, maintain 5 × 7 proportion

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